TESTING LUNAR RETURN THERMAL PROTECTION SYSTEMS USING SUB-SCALE FLIGHT TEST VEHICLES

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ABSTRACT

A key objective of NASA's Vision for Space Exploration is to revisit the lunar surface. Such an ambitious goal requires the development of a new human-rated spacecraft, the Orion Crew Exploration Vehicle (CEV), to ferry crews to low earth orbit and to the moon. The successful conclusion of both types of missions will require a thermal protection system (TPS) capable of protecting the vehicle and crew from the extreme heat of atmospheric reentry.

As a part of the TPS development, various materials are being tested in arcjet tunnels; however, the combined lunar return aerothermal environment of high heat flux, shear stress, and surface pressure cannot be duplicated using only existing ground test facilities. To ensure full TPS qualification, a flight test program using sub-scale Orion capsules has been proposed to test TPS materials and heat shield construction techniques under the most stressing combination of lunar return aerothermal environments. Originally called Testing Of Reentry Capsule Heat Shield, or TORCH, but later renamed LE-X, for Lunar Reentry Experiment, the proposed flight test program is presented along with the driving requirements and descriptions of the vehicle and the TPS instrumentation suite slated to conduct in-flight measurements.

1. RATIONALE & OVERVIEW

The development of ablative TPS materials, similar to those used on the Apollo command modules, virtually ceased following the early 1970's in favor of reusable tiles that are currently used on the Space Shuttle. While Shuttle tiles are perfectly suited for the aerothermal environments experienced during return from low Earth orbit, they are not capable of handling the environments the Orion heat shield will experience on return from the moon for which ablative TPS materials are required. Given the lack of development of ablative materials over the past 40 years, the agency finds itself without any efficient, high technology readiness level (TRL) options for the Orion heat shield. The Apollo Avocoat, the only lunar return qualified TPS material, has been out of production for decades and can no longer be considered as high TRL. As a result, NASA has initiated an advanced development project (ADP) to raise the TRL of several candidate ablative TPS materials through ground testing, analysis, and an assessment of manufacturing, repair, and operability risks.

Figure 1: LE-X at Atmospheric Interface

As will be discussed in the next section, testing of TPS materials under combined lunar-return aerothermal conditions in existing arcjet facilities will be problematic. Furthermore, verification of heat shield manufacturing techniques, such as segmented construction, will require test articles too large to be accommodated in existing or foreseeable arcjet test facilities. The Orion TPS ADP recognized the need to conduct flight tests similar to FIRE (Flight Investigation of Reentry Environments), performed during the Apollo-era [1]. This proposed Orion TPS flight test program was originally named TORCH, for Testing Of Reentry Capsule Heat Shield.

As the conceptual design of TORCH matured, it was soon noted within NASA that the Orion guidance, navigation, and control (GN&C) team was also proposing a sub-scale flight test for the purpose of testing the entry guidance. The GN&C flight test was referred to as "Yuma", in reference to the proposed landing site in Yuma, Arizona. Since both TORCH and Yuma were envisioned to be approximately two-fifth scale Orion capsules testing vehicle performance during atmospheric reentry, the two efforts were combined into a single project named **L**unar re**E**ntry e**X**periment, or LE-X (Fig. 1).

This paper presents design highlights of the LE-X spacecraft along with a discussion of the design trades that led to a cost effective flight system design, versatile enough to test the edges of the TPS flight envelope as well as other Orion subsystems. Also included is an overview of the trajectory design methodologies used to ensure that a sub-scale vehicle can achieve the same combined aerothermal conditions as the full scale Orion capsule. Finally, launch vehicle options for the various flight test missions are discussed.

Note that this paper will focus on the TPS testing objectives, although the skip entry guidance objectives will be briefly discussed in Section 3. As of this writing the process of merging the best aspects of the TORCH and Yuma designs into LE-X is still incomplete. This paper represents a snapshot of the design at the end of the TORCH preliminary study effort and, as such, is biased towards the TORCH configuration. The spacecraft is expected to evolve towards a unified LE-X design in the coming months.

2. AEROTHERMAL TEST OBJECTIVES

The combined aerothermal conditions experienced by an Orion vehicle during reentry are shown in Fig. 2. The blue curves represent expected environmental envelopes during return from the International Space Station (ISS). The dashed blue line represents the fully margined envelope during an actively guided reentry, the dotted line represents the envelope for a ballistic entry without margins, and the solid blue line is the fully margined ISS return envelope. Similarly, the red dashed, dotted, and solid lines represent the guided, unmargined ballistic, and fully margined ballistic envelopes, respectively, for lunar direct return (LDR).

The requirement to test at the extremes of the fully margined ballistic lunar return envelope (solid red line) drives the aerothermal requirements for LE-X as represented by the two red circular regions in Fig. 2. These two test regions, the ballistic heating and the ballistic shear conditions, have combined aerothermal conditions of total heat flux, radiative heat flux, and pressure as shown on the right side of Fig. 2.

The capabilities of existing and funded upgrades to ground test facilities is represented by the gray shaded region in Fig. 2. The maximum shear condition is beyond the capabilities of existing and planned upgrades. Although the maximum heat flux and radiative heating zone can be obtained in the upgraded arcjet facilities, the diminutive size of the test sections would preclude tests involving large, segmented TPS

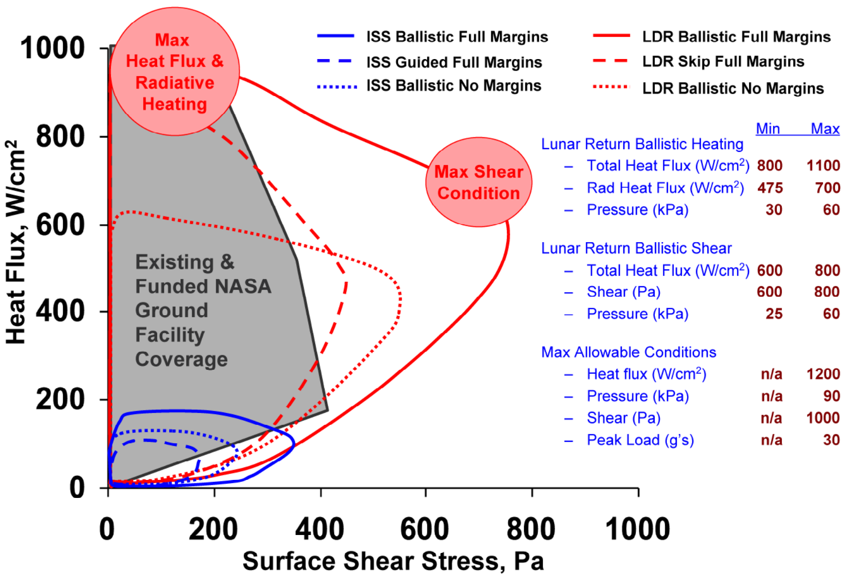


Figure 2: Required Aerothermal Test Conditions vs. Ground Test Facilities

panels. Testing of segmented heat shields is considered necessary since manufacturing limitations in the production of phenolic impregnated carbon ablator (PICA), the baselined Orion heat shield material, limits the maximum dimension of PICA panels to just over 1 m, thus requiring a tiled heat shield construction for the 5 m diameter spacecraft.

To avoid over-stressing the flight article, possibly resulting in vehicle failure and unfairly implicating the TPS, maximum allowable flight conditions have been specified at the bottom right of Fig. 2.

3. SKIP ENTRY OBJECTIVES

In order to maximize flexibility in lunar departure time for earth return and to land on or near the continental United States, the Orion design may include the capability to execute a skip entry profile to extend the downrange distance from entry interface. To execute such an entry profile, the GN&C subsystem would use the vehicle's lift to perform a brief exo-atmospheric ballistic coast phase following the first entry. With this technique, the range from the first entry interface to the landing site can be up to about 10,000 km. Although this technique has clear benefits, the skip entry guidance system has not been demonstrated by an American spacecraft; although, the former Soviet Union did execute a successful skip entry with the Zond 7 spacecraft in 1969. One of the primary objectives of LE-X is to demonstrate a skip entry using a vehicle with similar lift-to-drag ratio and ballistic number as the full-scale Orion spacecraft.

Although baselined as a primary source of navigational data during lunar return, the acquisition and tracking of global positioning system (GPS) signals at lunar return velocities has not been demonstrated. The LE-X mission would provide an ideal opportunity for such a demonstration.

The skip-entry test will also confirm computational fluid dynamics models describing interactions between the hypersonic aerodynamics and the attitude control thrusters used by the guidance algorithm to bank the vehicle for the purpose of re-orienting the lift vector.

4. BALANCING COSTS AND RISKS

Throughout the LE-X conceptual design phase, the flight test study team was focused on minimizing the overall project costs while maintaining an acceptable risk posture. The team concentrated on using small, lightweight, and simple test vehicles with nearly identical vehicle design for all test flights. The smaller and lighter the entry vehicle, the smaller the required launch vehicle, which usually translates to lower launch costs. Launch costs have historically been a substantial fraction of overall flight project costs. Additionally, the study team attempted to minimize the number of vehicle types in the project, thus reducing the non-recurring costs. A single design that could meet all the flight test objectives was desired even if the design was sub-optimal for a given objective.

For the sake of reducing costs and risks, the LE-X vehicle was envisioned as a simple, single-string vehicle without translational delta-v capability. With a mission duration measured in hours rather than years, dual-string vehicles are not necessary. Many of the key components of the vehicle that are often of lifetime concern such as the avionics, GN&C hardware, and telecommunications systems are of the same heritage as those of deep space vehicles designed to operate for years. The only deliberately added functional redundancy in the LE-X design is in the data acquisition and transmission system. Because acquisition and transmission of the on-board instrumentation data is arguably the most important requirement of the mission, those data are acquired on the ground by three independent ways, any one of which would be sufficient for mission success: 1) during hypersonic entry, flight telemetry from the TPS sensors and the GN&C system are transmitted to TDRSS (Tracking and Data Relay Satellite System) satellites in real-time and relayed to the ground, 2) during descent on the parachute, the accumulated telemetry is re-transmitted directly to ground stations near the landing site, and finally, 3) all data are stored on-board in non-volatile memory within the avionics so they can be retrieved when the capsule is recovered. The vehicle is further simplified by not requiring it to perform translational maneuvers to target the entry. The upper stage of the launch vehicle is required to place the on the desired flight trajectory.

One obvious way to minimize the project cost is to minimize the number of vehicles needed to be built and flown and still meet all the flight test objectives. To accomplish this goal, the mission design team attempted to maximize the number of test objectives that can be accomplished per flight. This will be discussed in the next section.

One of the early rules that the study team adopted was to avoid multiple test vehicles on a single launcher. Although this restriction may rule out some cost reduction options, the team believed strongly that the lessons from each flight should benefit the future ones. Furthermore, there was the desire to eliminate the possibility that one launch vehicle failure destroys two flight test vehicles.

Another early decision of the study team was the adoption of a 2/5 scale Orion capsule shape as the flight test vehicle's outer mold line. Although the use of alternate shapes may have allowed tailoring of the aerothermal conditions at various points on the vehicle, it was judged that there were additional analytical risks when extrapolating the test results of a sub-scale test vehicle to the real Orion vehicle if the two were of drastically different shapes. Additionally, the only aerothermal database currently available for LE-X analysis is a scaled derivative of the computational fluid dynamics (CFD) analysis of the full-scale Orion vehicle. A LE-X-specific CFD model and aerothermal database will be developed in preparation for detailed design.

5. MISSION DESIGN TRADE SPACE

The LE-X trade space was quite large so as to not eliminate the discovery of potential low cost solutions. Table 1 presents the full set of trade space parameters and associated ranges covered throughout the life cycle of the project. Trade space solutions were explored by parametrically varying key vehicle parameters with the remaining parameters being computed to produce the lowest cost test flight while satisfying the aerothermal and Guidance, Navigation, and Control (GN&C) test objectives. The key vehicle parameters that were varied were diameter and mass with typically the lowest associated entry velocity defining the lowest cost solution.

Table 1: Trade Space Parameters and Ranges

Parameters	Analysis Range
Mass (kg)	100 to 1450
Diameter (m)	0.5 to 4.0
Trim Angle of Attack (deg)	140 to 180
Bank Angle (deg)	0 to 180
Entry Flight Path Angle (deg)	4 to 20
Entry Speed (km/s)	5.5 to 12.1
Entry Type	Ballistic or Skip

To minimize mission costs, the study team tried to accomplish all the aerothermal and GN&C test objectives in the minimum number of flight test vehicles. The ultimate lowest cost mission design, if it were feasible, was, of course, a single test flight that performed a skip entry while also addressing both the aerothermal shear and heat flux/radiation objectives. The team examined options for two-flight programs, where one flight would be dedicated to executing the skip entry, and another would address only the aerothermal objectives. Some two-flight options combined the skip entry flight with one of the two aerothermal objectives. Finally, three-flight test programs were studied where there would be a dedicated skip entry flight as well, but the aerothermal objectives would be accomplished over two additional flights, one to hit the shear conditions and another for the heat flux/radiation condition. In summary, the seven mission design combinations explored were:

- Skip entry trajectory achieving the heat flux/radiation and shear conditions on the same flight
- Skip entry only trajectory
- Skip entry trajectory: heat flux/radiation condition only
- Skip entry trajectory: shear condition only
- Non-skip entry trajectory: heat flux/radiation and shear conditions on the same flight
- Non-skip entry trajectory: heat flux/radiation condition only
- Non-skip entry trajectory: shear condition only

NASA's Johnson Space Center (JSC) designed the skip entry test flight such that the sub-scale test vehicle would emulate, to the maximum extent possible, the GN&C interactions of the full scale CEV flight hardware and the environment. This required matching control response modes, the overall ballistic coefficient, and a trajectory that represented a skip entry mission design (i.e. the key objectives). The skip entry trade space effort was performed largely before any aerothermal requirements were introduced. Therefore, the aerothermal test objectives were evaluated against the mature skip entry vehicle configuration and mission design (i.e. much less flexibility was exercised in trying to meet the aerothermal objectives on the skip entry flight due to its maturity).

From the standpoint of achieving the aerothermal test conditions with minimal costs, the mission trades were suggesting smaller vehicle diameters. However, additional considerations drove the design to larger vehicles. In order to effectively test a segmented heat shield construction, the vehicle had to be large enough to test realistic tile seam running lengths. Furthermore, the vehicle size had to be large enough to trip turbulence in the aerodynamic flow around the vehicle and cause augmented heating and shear due to turbulence as is expected on the full-scale Orion capsule. The aggregate of all these considerations drove the selection of 2 m as the diameter of the LE-X vehicle.

It was quickly determined that the skip entry trajectory mission design could not meet the shear objective due to the fact that the altitude necessary to meet the shear condition was too low in the atmosphere to allow the vehicle's lifting force to propel it back into space sufficiently to perform the skip maneuver. However given enough entry velocity, the skip trajectory could meet the heat flux/radiation condition. The non-skip trajectory was also able to meet all aerothermal objectives on a single test flight and therefore, by definition, would be able to meet the heat flux/radiation and shear conditions on individual test flights if so desired. Ultimately, it was determined that two flight tests would represent the lowest cost test flight program: one dedicated for the GN&C objectives using a skip entry mission design and one dedicated to the aerothermal objectives (both heat flux/radiation and shear). The entry states associated with each flight test are presented in Table 2 along with other key parameters.

Table 2: Vehicle Entry States and Key Parameters

Parameter	Aerothermal Mission	Skip Entry Mission
Entry Mass (kg)	850	1250
Entry Speed (km/s)	12.1	~ 10
Entry Flight Path Angle (deg)	-6.71	~ -5
Diameter (m)	2	2
Trim Angle of Attack (deg)	162	156.8
Bank Angle (deg)	100 (fixed)	Guided

6. MISSION DESCRIPTION

Each of the two LE-X flights is currently envisioned to be sub-orbital, launching from the Kennedy Space Center and landing at the Woomera Test Facility in southern Australia. Consideration was given toward minimizing over flights of densely populated areas during atmospheric flight. Since Woomera is on the southern coast of Australia, an effort was made to approach from the south to keep most of the entry over the ocean. A so-called "pile-drive" trajectory was selected whereby the final launch vehicle stage is used to accelerate the LE-X capsule downward into the atmosphere to achieve the desired entry speed. This type of trajectory was chosen to reduce the apogee and, therefore, the flight time of the mission, thereby reducing the requirements on the power and thermal control subsystems. Although the pile-drive trajectory will require some non-standard analysis and simulation by the launch vehicle and mission design teams, a similar trajectory was successfully used on the FIRE Program from 1964-1965. Once the vehicle is subsonic over the landing site, a parachute will be used to reduce the touchdown speed to maximize preservation of the heat shield for analysis.

7. LAUNCH VEHICLE OPTIONS

Integral to the trade space effort was the launch vehicle pairings with the two flight tests. The launch vehicles considered are listed in Table 3 along with their compatibility with each flight test. Of the launch vehicles considered, the Minotaur family of launch vehicles built by Orbital Sciences Corporation (OSC) was eliminated as possible pairings with the two flight tests. All other launch vehicles remained in the trade space and, as of the writing of this paper, are still under consideration.

Table 3: Entry Vehicle Mass Capability by Mission

Parameter	Aerothermal Mission	Skip Entry Mission
OSC Minotaur IV	No Solution	No Solution
SpaceX Falcon 9	1300 kg	2550 kg
OSC Minotaur V	No Solution	No Solution
Boeing Delta II 7925H	No Solution	>1250 kg
LMA Atlas V 431	>850 kg	>1250 kg

8. FLIGHT SYSTEM DESCRIPTION

As illustrated in Fig. 3, each LE-X capsule is a 2-meter diameter, geometrically scaled model of the Orion crew module with a fully functional hypersonic guidance and control system. Where possible, the vehicle components and subsystems are based on flight-proven elements to minimize costs and risks.

The avionics subsystem is a copy of the Multi-mission System Architecture Platform (MSAP), the NASA Jet Propulsion Laboratory's institutional avionics suite slated for use in the Mars Science Laboratory, scheduled for launch in 2009. The MSAP suite contains a RAD750 flight computer, a 4 GB non-volatile memory card, a telecommunications interface card, peripheral interface electronics, pyro firing electronics, thruster drivers, and power switching and distribution electronics. Core flight software and operating system is available for the MSAP so that the LE-X team will only need to develop the mission-specific flight software.

The short flight duration of LE-X missions makes it possible to power the spacecraft in flight with batteries only, without the additional complexity of power generation equipment, such as solar arrays. The study team selected the rechargeable batteries used on the Mars Exploration Rover, which, as of this writing, have had over a thousand charge-discharge cycles on the surface of Mars. On the main equipment deck of the LE-X capsule there are five battery strings which can power the vehicle for approximately 3.5 hours, sufficient for the aerothermal flight. Because the skip entry flight may have durations as long as 15 hours, additional batteries can be included on the Extended Mission Deck as shown in Fig. 3.

The LE-X thermal design is a passive system using wax capacitors to store waste heat. Sufficient numbers of wax capacitors are distributed on the main equipment deck to keep the equipment at flight allowable temperatures during the aerothermal mission.

Additional wax capacitors are also included on the extended mission deck to permit longer flight times.

The LE-X spacecraft uses an S-Band telecommunications system to transmit telemetry to TDRSS satellites or to ground stations. Although no command uplink is planned during flight, the telecom system has the capability to accommodate it. There are two patch antennas located on the backshell and a helical antenna at the apex of the backshell. Antenna selection will be made by on-board logic to maximize coverage of the relay satellite or ground station, depending on mission phase.

The GN&C subsystem utilizes much of the same hardware as the real Orion vehicle so that the skip entry demonstration is as realistic as possible. The inertial measurement unit and GPS receiver will be the same type used on Orion. Furthermore, the reaction control system thrusters are configured similarly to that of Orion so that any potential interactions between aerodynamics and the thruster firings can be fully explored.

Each entry vehicle is contained within a highly instrumented aeroshell, traceable in design to the full-scale Orion capsule. The heat shield and backshell contain over 200 sensors positioned at key locations to optimize collection of TPS performance and aerothermal data. The TPS sensor suite includes thermocouples, recession sensors, pressure transducers, calorimeters, catalycity sensors, two spectrometers, two radiometers, and a radio frequency attenuation experiment. Data gathered by these instruments will be important for the certification of the full-scale Orion TPS.

9. RECOMMENDED FLIGHT TEST PROGRAM

The recommended LE-X flight test program, summarized in Fig. 4, consists of two flights using effectively identical copies of the same flight test vehicle. The skip entry vehicle will have an additional Extended Mission Deck to support a longer flight time, as was discussed in Sect. 8, and additional ballast to match the ballistic number of the full-scale Orion vehicle.

The Mars Exploration Rover experience has shown that when identical spacecraft are built concurrently, using the same personnel, procedures, and tooling, the marginal cost of building additional vehicles can be very low. To minimize the project costs it is important that the vehicles are constructed in parallel, because even a short gap between builds may require retraining of personnel, and could result in parts availability issues and obsolescence of equipment.

As a risk reduction measure, the study team is recommending simultaneous procurement of all parts for two vehicles and long-lead (>6 months) components for a third. This approach will allow rapid fabrication of third vehicle in case of a mission failure on one flight.

The selection of launch vehicles has not been determined as of this writing. Although flight-proven commercial launchers such as the Lockheed-Martin Atlas V and the Boeing Delta II-H can deliver the LE-X vehicles to the desired entry conditions, the SpaceX Falcon 9, currently under development, can also do so at a considerably lower cost. Since the proposed LE-X project is still at least two years away from selecting a launch vehicle, the design is still keeping all launch vehicle options open at this time. The proposed first flight date is between 2012 and 2014.

10. CONCLUSIONS

The potential inability to certify the Orion heat shield for lunar return is one of the greatest threats to NASA's Vision for Space Exploration. The LE-X study team has shown that a cost effective program to retire the risks associated with the Orion lunar return thermal protection system and skip entry guidance is possible using sub-scale flight test vehicles. By using heritage components and techniques leveraged from the robotic planetary space program, such a flight test program will likely be less expensive than flying a full-scale but uncrewed Orion vehicle on a lunar-return trajectory.

The LE-X design and infrastructure can also be leveraged for future flight test objectives. Future copies of LE-X capsules can be built and flown to support research into high-speed flows. LE-X vehicles can also be modified to support development of future interplanetary mission, such as the testing of Mars return TPS materials or demonstrating aerocapture.

11. REFERENCES

1. General Dynamics Convair Report No. GDC/BKF65-042, "Project FIRE Integrated Post Flight Evaluation Report, Flight II", 24 September 1965. (NASA CR66000)

The research described in this paper was carried out at the Ames Research Center; Jet Propulsion Laboratory, California Institute of Technology; Johnson Space Center; Langley Research Center; and Sandia National Laboratories under a contract with the National Aeronautics and Space Administration.

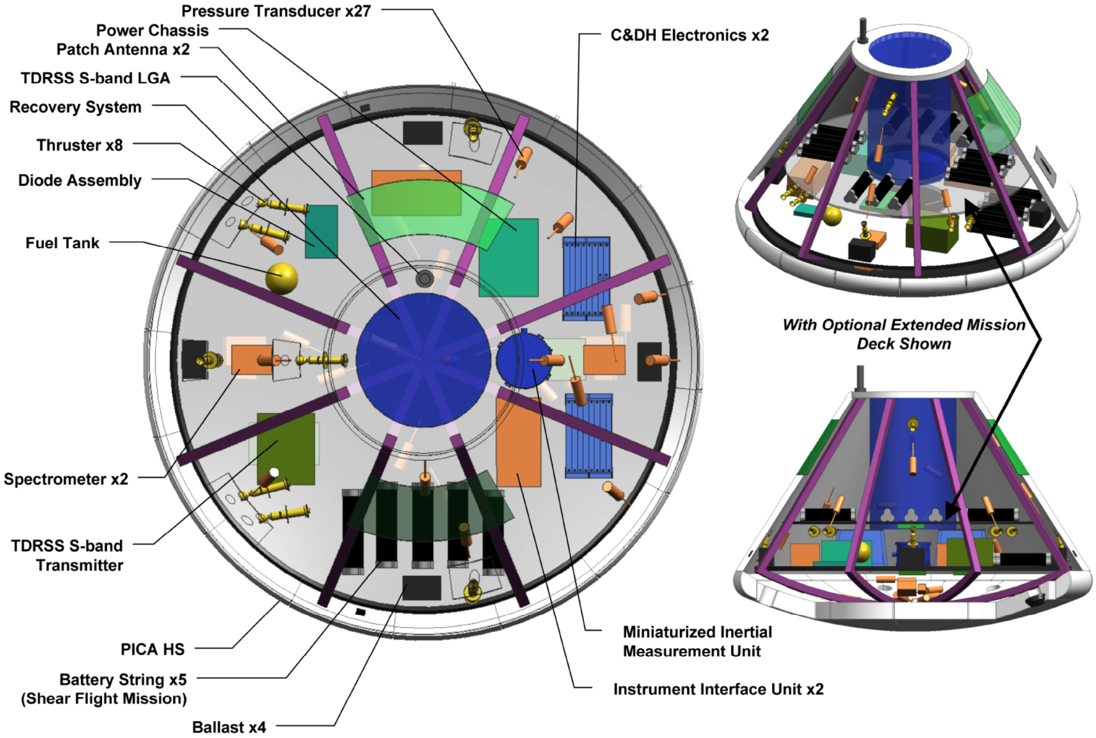
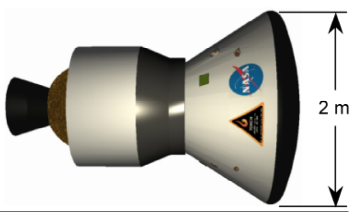


Figure 3 : LE-X Vehicle Layout



	Atlas V / Falcon 9 Class LDR Ballistic	Delta II / Falcon 9 Class LDR Skip	Program Total
Entry Vehicle Diameter (m)	2	2	
Entry Vehicle Mass (kg)	850	1250	
Inertial Entry Velocity (km/sec)	12.1	10.5	
Inertial Entry Flight Path Angle (deg)	-6.7	-6.3	
Flight Vehicle (2 with long lead parts for 3rd), Instrumentation, & Operations Cost / DDT&E	$103M	$36M	$139M
Flight Cost excluding Launch Vehicle (w/ reserves applied to both vehicles)*	$122M (incl. 1st flight dev. costs)	$43M (recurring cost of 2nd flight test vehicle only)	$165M

*** Flight Costs:**

- **Costs in FY07 dollars without inflation**
- **Assumes two identical and concurrent flight builds.**

- Two launches recommended from both a technical and risk perspective.
- For two launches, expected launch costs range are $70M-$250M.
- Launch costs may be less as there is the possibility that they have already been covered by the Agency.

Figure 4 : Recommended LE-X Flight Test Program